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(54) **COMBUSTER WITH RADIAL FUEL INJECTION**

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(51) **Int. Cl.**

F23R 3/28	(2006.01)
F23R 3/34	(2006.01)
F23K 5/20	(2006.01)
F23R 3/44	(2006.01)

(52) **U.S. Cl.**

CPC ... ***F23R 3/28*** (2013.01); ***F23K 5/20*** (2013.01); ***F23R 3/34*** (2013.01); ***F23R 3/346*** (2013.01); ***F23R 3/44*** (2013.01)

(58) **Field of Classification Search**

CPC F23R 3/28; F23R 3/346; F23R 3/44; F23R 3/34; F23R 3/343; F23K 5/20; F23K 2301/204

USPC 60/734, 739, 746, 733, 736

See application file for complete search history.

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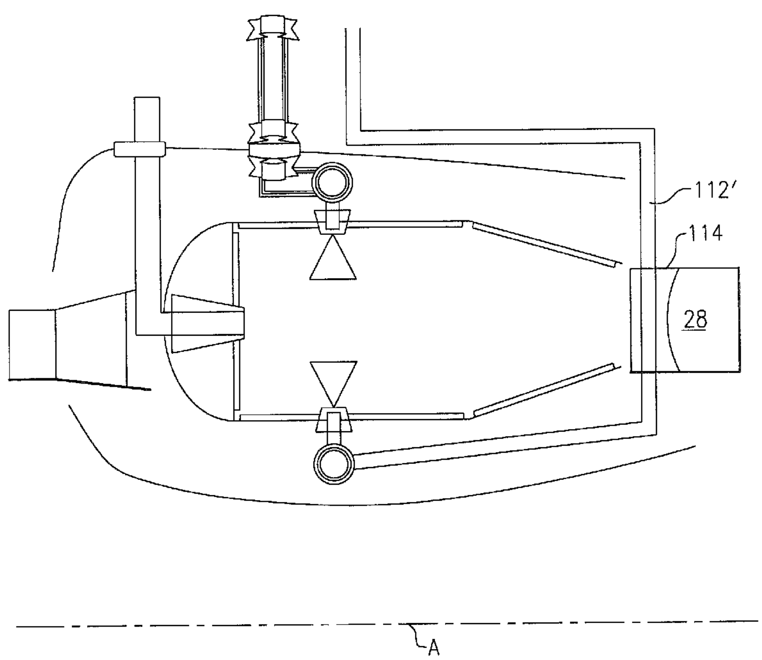
(57) **ABSTRACT**

A combustor for a gas turbine engine includes an forward fuel injection system in communication with a combustion chamber and a downstream fuel injection system that communicates with the combustion chamber downstream of the forward fuel injection system.

9 Claims, 4 Drawing Sheets



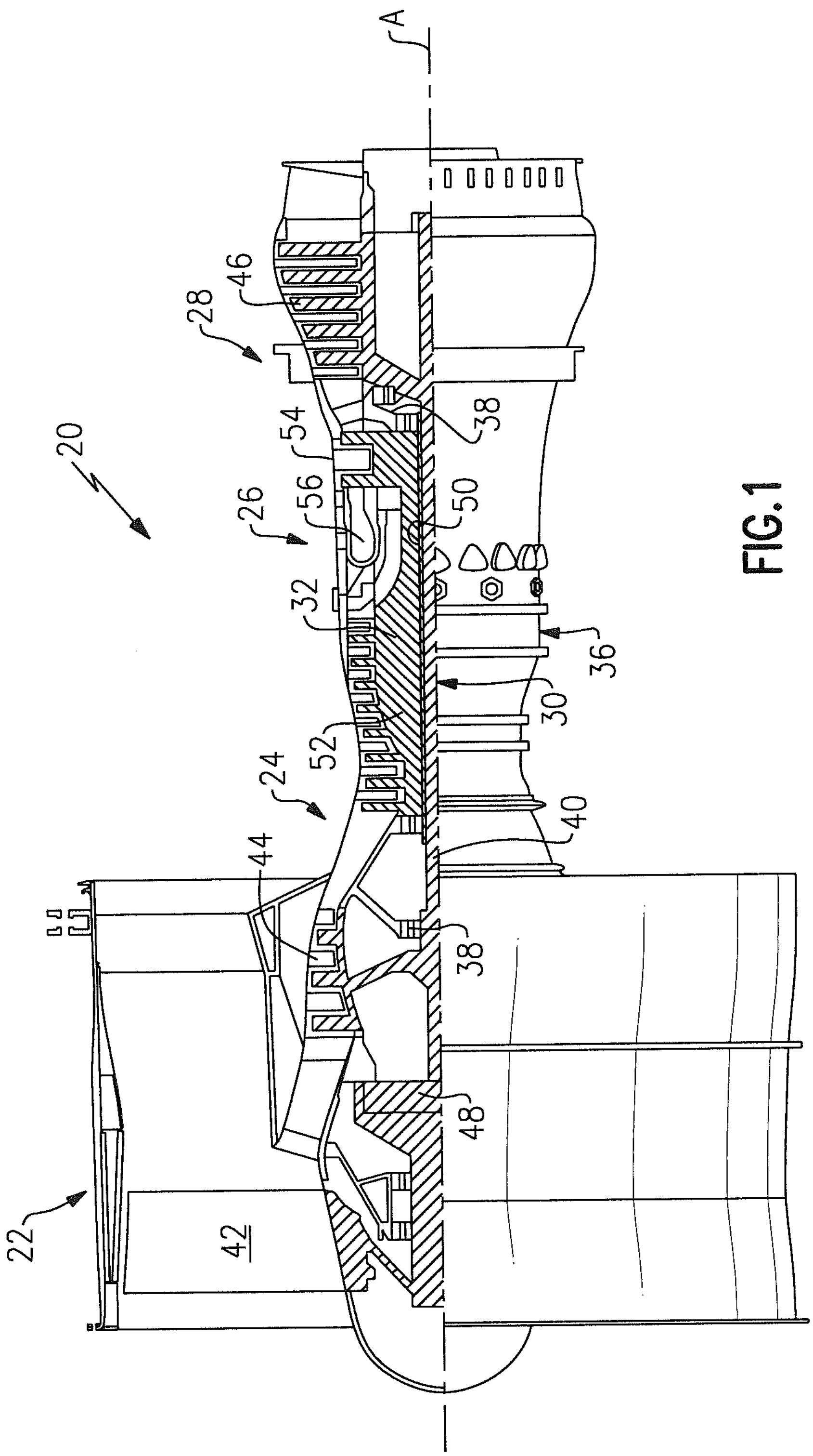
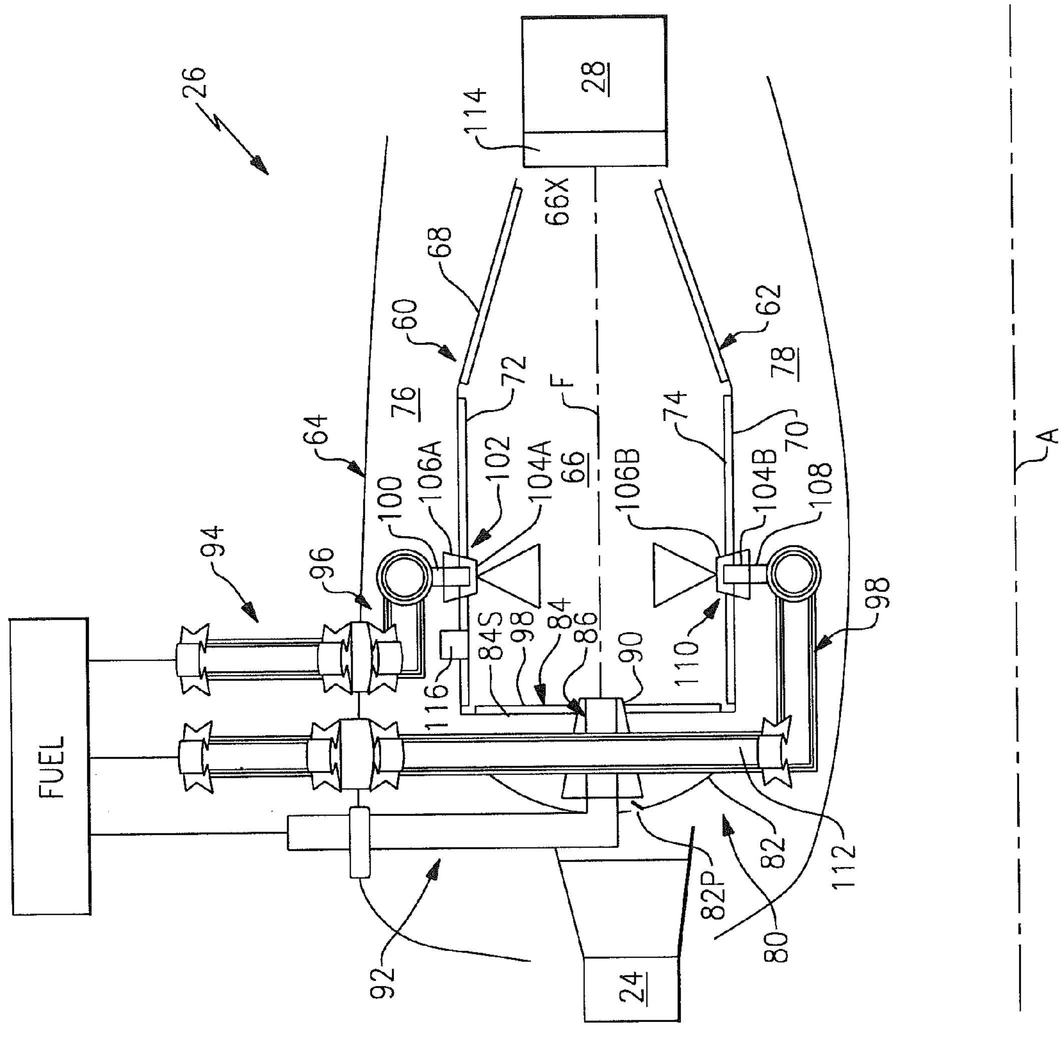


FIG. 1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104B
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FIG.2

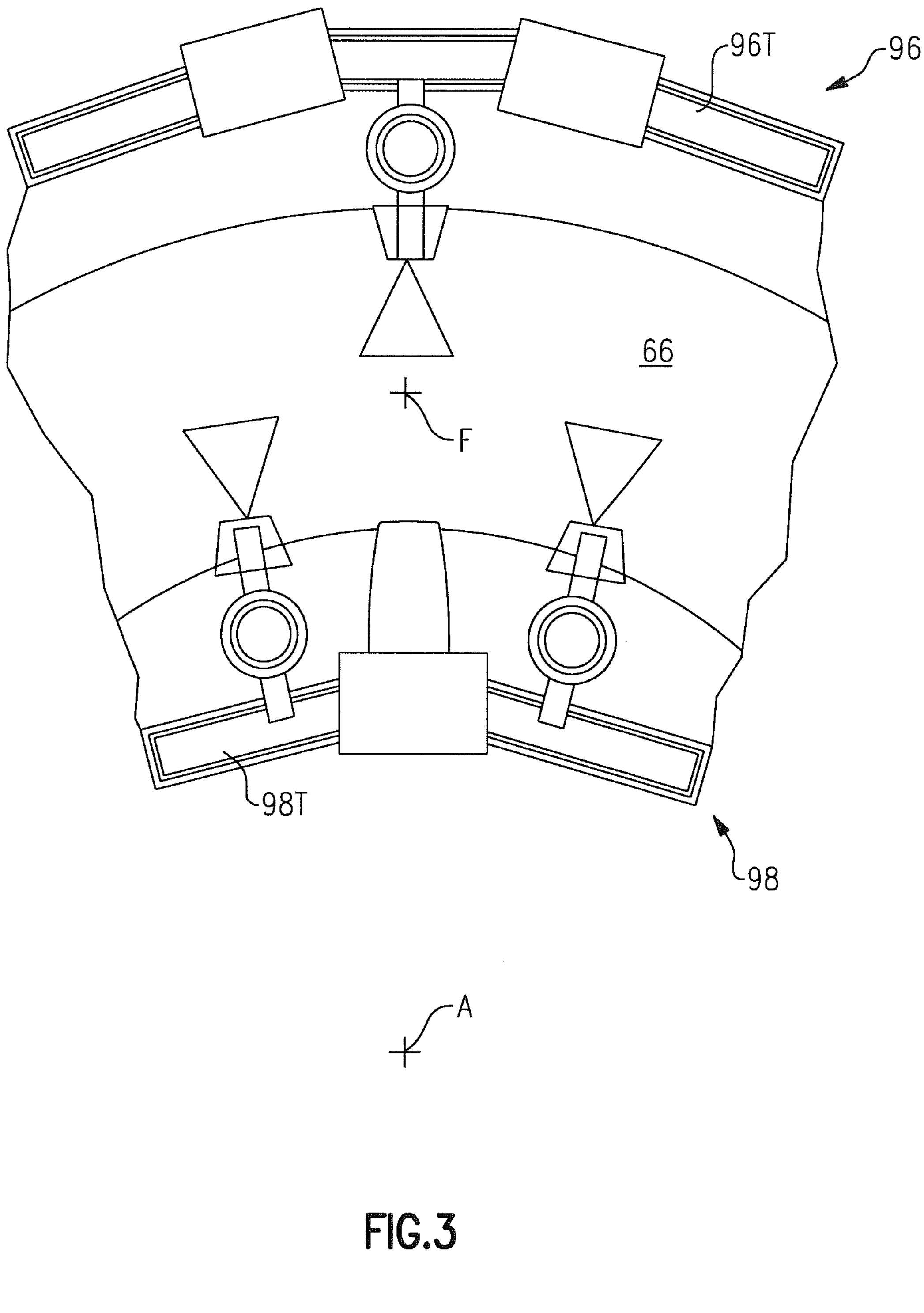

FIG.3

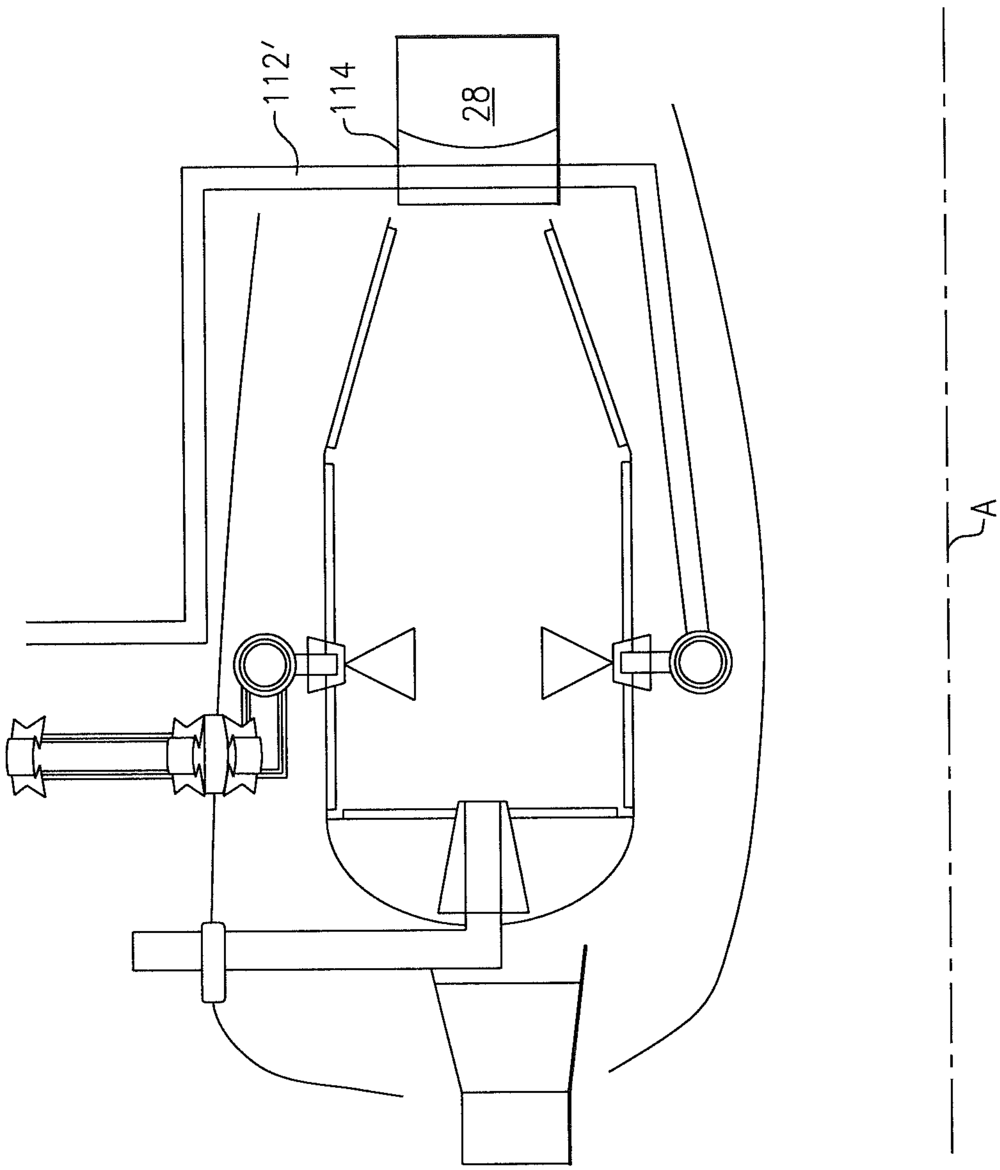

FIG.4

COMBUSTER WITH RADIAL FUEL INJECTION

Applicant hereby claims priority to U.S. Patent Application No. 61/707,033 filed Sep. 28, 2012, the disclosure of which is herein incorporated by reference.

BACKGROUND

The present disclosure relates to a gas turbine engine and, more particularly, to a fuel nozzle arrangement therefor.

Gas turbine engines, such as those which power modern commercial and military aircraft, include a compressor for pressurizing a supply of air, a combustor for burning a hydrocarbon fuel in the presence of the pressurized air, and a turbine for extracting energy from the resultant combustion gases. The combustor generally includes radially spaced apart inner and outer liners that define an annular combustion chamber therebetween. Arrays of circumferentially distributed combustion air holes penetrate multiple axial locations along each liner to radially admit the pressurized air into the combustion chamber. A plurality of circumferentially distributed fuel injectors axially project into a forward section of the combustion chamber to supply the fuel for mixing with the pressurized air.

Combustion of the hydrocarbon fuel in the presence of pressurized air may produce nitrogen oxide (NO_x) emissions that are subject to excessively stringent controls by regulatory authorities, and thus may be sought to be minimized as much as possible.

At least one known strategy for minimizing NO_x emissions is referred to as rich burn, quick quench, lean burn (RQL) combustion. The RQL strategy recognizes that the conditions for NO_x formation are most favorable at elevated combustion flame temperatures, such as when a fuel-air ratio is at or near stoichiometric. A combustor configured for RQL combustion includes three serially arranged combustion zones: a Rich burn zone at the forward end of the combustor, a Quench or dilution zone axially aft of the rich burn zone, and a Lean burn zone axially aft of the quench zone.

During engine operation, a portion of the pressurized air discharged from the compressor enters the rich burn zone of the combustion chamber. Concurrently, the fuel injectors introduce a stoichiometrically excessive quantity of fuel into the rich burn zone. Although the resulting stoichiometrically fuel rich fuel-air mixture is ignited and burned to partially release the energy content of the fuel, NO_x formation may still occur.

The fuel rich combustion products then enter the quench zone where jets of pressurized air radially enter through combustion air holes from the compressor and into the quench zone of the combustion chamber. The pressurized air mixes with the combustion products to support further combustion of the fuel with air by progressively deriching the fuel rich combustion products as they flow axially through the quench zone and mix with the air. Initially, the fuel-air ratio of the combustion products changes from fuel rich to stoichiometric, causing an attendant rise in the combustion flame temperature. Since the quantity of NO_x produced in a given time interval is known to increase exponentially with flame temperature, quantities of NO_x may be produced during the initial quench process. As the quenching continues, the fuel-air ratio of the combustion products changes from stoichiometric to fuel lean, causing an attendant reduction in the flame temperature. However, until the mixture is diluted to a fuel-air ratio substantially lower than stoichiometric, the flame temperature remains high enough to generate NO_x.

Finally, the deriched combustion products from the quench zone flow axially into the lean burn zone. Additional pressurized air in this zone supports ongoing combustion to release energy from the fuel. The additional pressurized air in this zone also regulates the peak temperature and spatial temperature profile of the combustion products to reduce turbine exposure to excessive temperatures and excessive temperature gradients.

SUMMARY

A combustor for a gas turbine engine according to one disclosed non-limiting embodiment of the present disclosure includes an forward fuel injection system in communication with a combustion chamber, and a downstream fuel injection system that communicates with said combustion chamber downstream of said forward fuel injection system.

In a further embodiment of the foregoing embodiment, the downstream fuel injection system at least partially surrounds the combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially inboard of the combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially outboard of the combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially outboard and radially inboard of the combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system includes a multiple of fuel nozzle assemblies axially upstream of a necked region of the combustor.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system includes a multiple of fuel nozzle assemblies within a first two-thirds of the combustor.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially inboard of the combustion chamber, a main supply line of a radially inner fuel injection manifold extends through a forward assembly.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially inboard of the combustion chamber, a main supply line of a radially inner fuel injection manifold extends through a downstream vane

A gas turbine engine according to another disclosed non-limiting embodiment of the present disclosure includes an forward fuel injection system in communication with a combustion chamber and a downstream fuel injection system around said combustion chamber, said downstream fuel injection system communicates with said combustion chamber downstream of said forward fuel injection system.

In a further embodiment of the foregoing embodiment, the downstream fuel injection system is radially inboard of said combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially outboard of said combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system is radially outboard and radially inboard of said combustion chamber.

In a further embodiment of any of the foregoing embodiments, the downstream fuel injection system includes a multiple of fuel nozzle assemblies axially upstream of a necked region of said combustor. In the alternative or additionally thereto, in the foregoing embodiment the downstream fuel injection system includes a multiple of fuel nozzle assemblies within a first two-thirds of said combustor. In the alternative or additionally thereto, in the foregoing embodiment the downstream fuel injection system is radially inboard of said combustion chamber, a main supply line of a radially inner fuel injection manifold extends through a forward assembly. In the alternative or additionally thereto, in the foregoing embodiment the downstream fuel injection system is radially inboard of said combustion chamber, a main supply line of a radially inner fuel injection manifold extends through a downstream vane.

A method of communicating fuel to a combustor of a gas turbine engine, according to another disclosed non-limiting embodiment of the present disclosure includes communicating fuel axially into a combustion chamber and communicating fuel radially into the combustion chamber.

In a further embodiment of the foregoing embodiment, the method includes communicating fuel radially inward into the combustion chamber.

In a further embodiment of the foregoing embodiment, the method includes communicating fuel radially outward into the combustion chamber.

BRIEF DESCRIPTION OF THE DRAWINGS

Various features will become apparent to those skilled in the art from the following detailed description of the disclosed non-limiting embodiment. The drawings that accompany the detailed description can be briefly described as follows:

FIG. **1** is a schematic cross-section of a gas turbine engine;

FIG. **2** is a partial longitudinal schematic sectional view of an exemplary annular combustor that may be used with the gas turbine engine shown in FIG. **1**;

FIG. **3** is a partial lateral schematic sectional view of an exemplary annular combustor of FIG. **2**; and

FIG. **4** is a partial longitudinal schematic sectional view of an exemplary annular combustor according to another non-limiting embodiment, that may be used with the gas turbine engine shown in FIG. **1**.

DETAILED DESCRIPTION

FIG. **1** schematically illustrates a gas turbine engine **20**. The gas turbine engine **20** is disclosed herein as a two-spool turbofan that generally incorporates a fan section **22**, a compressor section **24**, a combustor section **26** and a turbine section **28**. Alternative engines might include an augmentor section (not shown) among other systems or features. The fan section **22** drives air along a bypass flowpath while the compressor section **24** drives air along a core flowpath for compression and communication into the combustor section **26** then expansion through the turbine section **28**. Although depicted as a turbofan gas turbine engine in the disclosed non-limiting embodiment, it should be understood that the concepts described herein are not limited to use with turbofans as the teachings may be applied to other types of turbine engines such as a three-spool (plus fan) engine wherein an intermediate spool includes an intermediate pressure compressor (IPC) between the LPC and HPC and an intermediate pressure turbine (IPT) between the HPT and LPT as well as aero-derivative/electrical power engine applications.

The engine **20** generally includes a low spool **30** and a high spool **32** mounted for rotation about an engine central longitudinal axis A relative to an engine static structure **36** via several bearing structures **38**. The low spool **30** generally includes an inner shaft **40** that interconnects a fan **42**, a low pressure compressor **44** ("LPC") and a low pressure turbine **46** ("LPT"). The inner shaft **40** drives the fan **42** directly or through a geared architecture **48** to drive the fan **42** at a lower speed than the low spool **30**. An exemplary reduction transmission is an epicyclic transmission, namely a planetary or star gear system.

The high spool **32** includes an outer shaft **50** that interconnects a high pressure compressor **52** ("HPC") and high pressure turbine **54** ("HPT"). A combustor **56** is arranged between the high pressure compressor **52** and the high pressure turbine **54**. The inner shaft **40** and the outer shaft **50** are concentric and rotate about the engine central longitudinal axis A which is collinear with their longitudinal axes.

Core airflow is compressed by the low pressure compressor **44** then the high pressure compressor **52**, mixed with the fuel and burned in the combustor **56**, then expanded over the high pressure turbine **54** and low pressure turbine **46**. The turbines **54**, **46** rotationally drive the respective low spool **30** and high spool **32** in response to the expansion.

The main engine shafts **40**, **50** are supported at a plurality of points by bearing structures **38** within the static structure **36**. It should be understood that various bearing structures **38** at various locations may alternatively or additionally be provided.

In one non-limiting example, the gas turbine engine **20** is a high-bypass geared aircraft engine. In a further example, the gas turbine engine **20** bypass ratio is greater than about six (6:1). The geared architecture **48** can include an epicyclic gear train, such as a planetary gear system or other gear system. The example epicyclic gear train has a gear reduction ratio of greater than about 2.3, and in another example is greater than about 2.5:1. The geared turbofan enables operation of the low spool **30** at higher speeds which can increase the operational efficiency of the low pressure compressor **44** and low pressure turbine **46** and render increased pressure in a fewer number of stages.

A pressure ratio associated with the low pressure turbine **46** is pressure measured prior to the inlet of the low pressure turbine **46** as related to the pressure at the outlet of the low pressure turbine **46** prior to an exhaust nozzle of the gas turbine engine **20**. In one non-limiting embodiment, the bypass ratio of the gas turbine engine **20** is greater than about ten (10:1), the fan diameter is significantly larger than that of the low pressure compressor **44**, and the low pressure turbine **46** has a pressure ratio that is greater than about 5 (5:1). It should be understood, however, that the above parameters are only exemplary of one embodiment of a geared architecture engine and that the present disclosure is applicable to other gas turbine engines including direct drive turbofans.

In one embodiment, a significant amount of thrust is provided by the bypass flow path B due to the high bypass ratio. The fan section **22** of the gas turbine engine **20** is designed for a particular flight condition—typically cruise at about 0.8 Mach and about 35,000 feet. This flight condition, with the gas turbine engine **20** at its best fuel consumption, is also known as bucket cruise Thrust Specific Fuel Consumption (TSFC). TSFC is an industry standard parameter of fuel consumption per unit of thrust.

Fan Pressure Ratio is the pressure ratio across a blade of the fan section **22** without the use of a Fan Exit Guide Vane system. The low Fan Pressure Ratio according to one non-limiting embodiment of the example gas turbine engine **20** is less than 1.45. Low Corrected Fan Tip Speed is the actual fan tip speed divided by an industry standard temperature correction of "T"/518.70.5. in which "T" represents the ambient temperature in degrees Rankine. The Low Corrected Fan Tip

Speed according to one non-limiting embodiment of the example gas turbine engine **20** is less than about 1150 fps (351 m/s).

With reference to FIG. **2**, the combustor **56** generally includes a combustor outer liner **60** and a combustor inner liner **62**. The outer liner **60** and the inner liner **62** are spaced inward from a diffuser case **64** such that a combustion chamber **66** is defined therebetween. The combustion chamber **66** is generally annular in shape and is defined between combustor liners **60**, **62**.

The outer liner **60** and the diffuser case **64** define an outer annular plenum **76** and the inner liner **62** and the case **64** define an inner annular plenum **78**. It should be understood that although a particular combustor is illustrated, other combustor types with various combustor liner panel arrangements will also benefit herefrom.

Each liner **60**, **62** generally includes a respective support shell **68**, **70** that supports one or more respective liner panels **72**, **74** mounted to a hot side of the respective support shell **68**, **70**. Each of the liner panels **72**, **74** may be generally rectilinear and manufactured of, for example, a nickel based super alloy or ceramic material.

The combustor **56** further includes a forward assembly **80** immediately downstream of the compressor section **24** to receive compressed airflow therefrom. The forward assembly **80** generally includes an annular hood **82**, a bulkhead assembly **84**, a multiple of axial fuel nozzles **86** (one shown; illustrated schematically) and a multiple of swirler assemblies **90** (one shown; illustrated schematically) that define a central opening. The annular hood **82** extends radially between, and is secured to, the forwardmost ends of the liners **60**, **62**. The annular hood **82** includes a multiple of circumferentially distributed hood ports **82**P that accommodate the respective fuel nozzle **86** and introduces air into the forward end of the combustion chamber **66**. The centerline of the fuel nozzle **86** is concurrent with the centerline F of the respective swirler assembly **90**. Each fuel nozzle **86** may be secured to the diffuser case **64** to project through one of the hood ports **82**P and through the central opening **90**A within the respective swirler assembly **90**. It should be understood that some combustors, such as lean or front-end staged combustors, may have more complex front end geometries in which fuel nozzles may be oriented other than in a circumferential pattern.

Each swirler assembly **90** is circumferentially aligned with, and/or concentric to, one of the hood ports **82**P to project through the bulkhead assembly **84**. Each bulkhead assembly **84** includes a bulkhead support shell **84**S secured to the liners **60**, **62**, and a multiple of circumferentially distributed bulkhead heatshields segments **98** secured to the bulkhead support shell **84**S around the central opening **90**A.

The forward assembly **80** directs a portion of the core airflow into the forward end of the combustion chamber **66** while the remainder enters the outer annular plenum **76** and the inner annular plenum **78**. The multiple of axial fuel nozzles **86**, swirler assemblies **90** and associated fuel communication structure defines a forward fuel injection system **92** that supports combustion in the combustion chamber **66**.

A downstream fuel injection system **94** communicates with the combustion chamber **66** downstream of the forward fuel injection system **92**. The downstream fuel injection system **94** introduces a portion of the fuel required for desired combustion performance, e.g., emissions, operability, durability as well as to lean-out the fuel contribution provided by the multiple of axial fuel nozzles **86** generally parallel to axis F.

The downstream fuel injection system **94** generally includes a radially outer fuel injection manifold **96** located in the outer annular plenum **76** and/or a radially inner fuel injection manifold **98** located in the inner annular plenum **78**. It should be appreciated that the downstream fuel injection system **94** may include only the radially outer fuel injection manifold **96**; only the radially inner fuel injection manifold **98** or both (shown).

The radially outer fuel injection manifold **96** may be mounted to the diffuser case **64**. Alternatively, the radially outer fuel injection manifold **96** may be mounted to the shell **68**. The radially inner fuel injection manifold **98** may be mounted to the diffuser case or shell **70**. It should be appreciated that various mount arrangements may alternatively or additionally provided such as location of the outer fuel injection manifold **96** mounted inside or outside the diffuser case **64**.

The radially outer fuel injection manifold **96** and the radially inner fuel injection manifold **98** may be manufactured of a series of straight tube sections **96**T, **98**T that may be connected together by a series of joints or fittings via braze or weld methods (FIG. **3**). It should be appreciated that various assembly methods and component structures may be alternatively or additionally be provided.

The radially outer fuel injection manifold **96** includes a multiple of radially extending supply lines **100** which terminate in an outer fuel nozzle assembly **102** that project predominantly radially toward the centerline F of the combustor chamber **66**. The multiple of radially extending supply lines **100** may include, for example, compliant fuel lines or pigtails that accommodate relative growth and part movement. In one disclosed non-limiting embodiment, the outer fuel nozzle assembly **102** includes fuel injector ports **104**A encased by an air swirler **106**A that promote mixing of the fuel spray with air from within the diffuser case **64** to facilitate generation of the fuel-air distribution required for combustion.

The radially inner fuel injection manifold **98** likewise includes a multiple of radially extending supply lines **108** which terminate in an inner fuel nozzle assembly **110** that project predominantly radially toward the centerline F of the combustor chamber **66**. The multiple of radially extending supply lines **108** may include, for example, compliant fuel lines or pigtails that accommodate relative growth and part movement. In one disclosed non-limiting embodiment, the inner fuel nozzle assembly **110** include fuel injector ports **104**B encased by an air swirler **106**B that promote mixing of the fuel spray with air from within the diffuser case **64** to facilitate generation of the fuel-air distribution required for combustion.

The radially inner fuel injection manifold **98** includes a main supply line **112** which may be arranged to pass through the relatively cooler forward assembly **80** to provide communication with the multiple of radially extending supply lines **108**. Alternatively, the main supply line **112** may pass through a downstream vane **114** such as a Nozzle Guide Vane (FIG. **4**). It should be appreciated that the main supply line **112** may be a secondary or intermediary fuel line to, for example, facilitate assembly.

Given operational temperatures from the HPC **52**, the radially outer fuel injection manifold **96** and the radially inner fuel injection manifold **98** may be subject to soaking temperatures that may promote coking. The radially outer fuel injection manifold **96** and the radially inner fuel injection manifold **98** and other associated lines may be configured with a protective, low-conductivity sheath, a coating, a cooled tube-in-tube construction, be relatively oversized compared to fuel flow or other insulation that provides thermal resistance between the relatively hot air temperatures in the diffuser case **64** and the relatively cold fuel temperatures in the fuel lines, manifolds and nozzles. Alternatively, or in addition, the downstream fuel injection system **94** may communicate through or with the bypass stream of the engine and may include a thermal management or heat exchange system to further maintain low fuel temperatures.

The outer and inner fuel nozzle assemblies **102**, **110** project through openings in the combustor **56** to supply fuel to the combustor between the bulkhead assembly **84** and a combustor exit **66***x*. In one disclosed non-limiting embodiment, the outer and inner fuel nozzle assemblies **102**, **110** project through openings in the combustor **56** located within the first two-thirds of the combustor chamber **66**. In another disclosed non-limiting embodiment, the outer and inner fuel nozzle assemblies **102**, **110** project through openings in the combustor **66** between 20-70% of the axial length. In another disclosed non-limiting embodiment, the outer and inner fuel nozzle assemblies **102**, **110** project through openings in the combustor **66** upstream of a necked region **56**N of the combustor **56**. That is, an internal height of the bulkhead assembly **84** is greater than the combustor exit **66***x*.

Spark energy may be provided to the combustor **56** through a frequency-pulsed igniter arrangement **116** (illustrated schematically) which provides a continuous spark or other ignition source. The frequency-pulsed igniter arrangement **116** may be located in conventional as well as other locations within the combustor **56**.

The fuel required for combustion is, thus, provided by the both the axial fuel nozzles **86** and the fuel nozzles **102**, **110** associated with the radially outer fuel injection manifold **96** and the radially inner fuel injection manifold **98**. The distributed fuel injection and fuel-air mixing provided thereby may be tailored to optimize emissions, e.g., NOx, COx, smoke, particulates, etc., as well as control of combustor thermals, durability, profile and pattern factors that impact the downstream turbine section.

It should be understood that relative positional terms such as “forward,” “aft,” “upper,” “lower,” “above,” “below,” and the like are with reference to the normal operational attitude of the vehicle and should not be considered otherwise limiting.

It should be understood that like reference numerals identify corresponding or similar elements throughout the several drawings. It should also be understood that although a particular component arrangement is disclosed in the illustrated embodiment, other arrangements will benefit herefrom.

Although particular step sequences are shown, described, and claimed, it should be understood that steps may be performed in any order, separated or combined unless otherwise indicated and will still benefit from the present disclosure.

The foregoing description is exemplary rather than defined by the limitations within. Various non-limiting embodiments are disclosed herein, however, one of ordinary skill in the art would recognize that various modifications and variations in light of the above teachings will fall within the scope of the appended claims. It is therefore to be understood that within the scope of the appended claims, the disclosure may be practiced other than as specifically described. For that reason the appended claims should be studied to determine true scope and content.

What is claimed is:

1. A combustor for a gas turbine engine comprising:

a forward fuel injection system in communication with a combustion chamber; and

a downstream fuel injection system that communicates with said combustion chamber downstream of said forward fuel injection system,

wherein said downstream fuel injection system is radially inboard of said combustion chamber and a main supply line of a radially inner fuel injection manifold extends through a vane in a turbine section downstream of said combustion chamber.

2. The combustor as recited in claim **1**, wherein said downstream fuel injection system at least partially surrounds said combustion chamber.

3. The combustor as recited in claim **1**, wherein said downstream fuel injection system is radially outboard of said combustion chamber.

4. The combustor as recited in claim **1**, wherein said downstream fuel injection system includes a multiple of fuel nozzle assemblies axially upstream of a necked region of said combustor.

5. The combustor as recited in claim **1**, wherein said downstream fuel injection system includes a multiple of fuel nozzle assemblies within a first two-thirds of said combustor.

6. A gas turbine engine comprising:

a forward fuel injection system in communication with a combustion chamber; and

a downstream fuel injection system at least partially around said combustion chamber, said downstream fuel injection system communicates with said combustion chamber downstream of said forward fuel injection system,

wherein said downstream fuel injection system is radially inboard of said combustion chamber and a main supply line of a radially inner fuel injection manifold extends through a vane in a turbine section downstream of said combustion chamber.

7. The gas turbine engine as recited in claim **6**, wherein said downstream fuel injection system is radially outboard of said combustion chamber.

8. The gas turbine engine as recited in claim **6**, wherein said downstream fuel injection system includes a multiple of fuel nozzle assemblies axially upstream of a necked region of said combustor.

9. The gas turbine engine as recited in claim **8**, wherein said downstream fuel injection system includes a multiple of fuel nozzle assemblies within a first two-thirds of said combustor.

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